Voyager Backgrounder

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Charles Redmond
NASA Headquarters, Washington, D.C.
(Phone: 202/755-3680)

Frank Bristow
NASA Jet Propulsion Laboratory, Pasadena, Calif.
(Phone: 213/354-5011)

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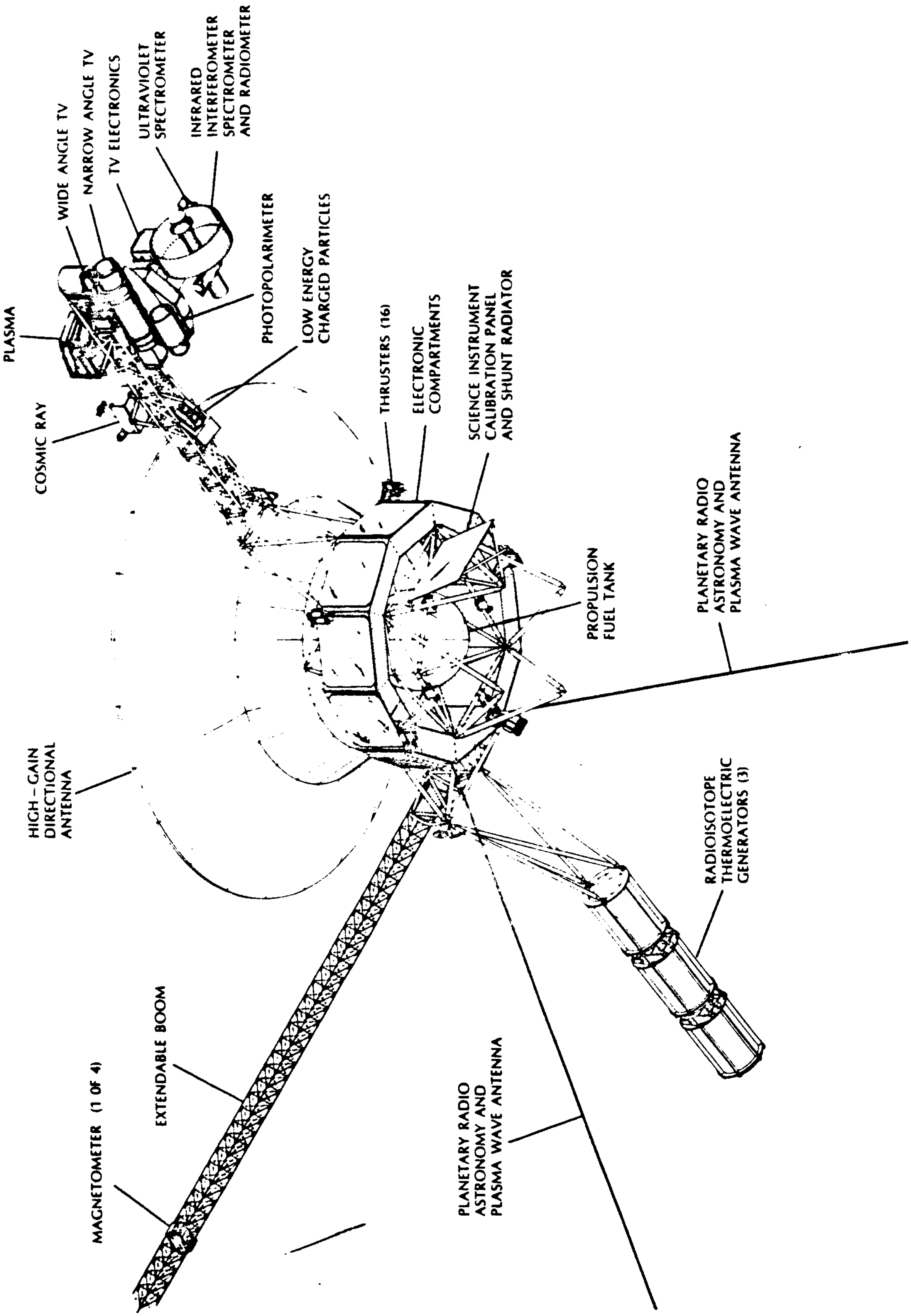
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THE VOYAGER SPACECRAFT

The two identical Voyager spacecraft are designed to operate at greater distances from Earth and the Sun than required by any previous mission. Communications capability, hardware reliability, navigation and temperature control are among the major challenges that had to be met to operate over the time of the mission, because of the Earth-spacecraft distances involved and the wide range of environmental conditions encountered.

Each spacecraft at launch consisted of a mission module -- the planetary vehicle -- and a propulsion module, which provided the final energy increment to inject the mission module onto the Jupiter trajectory. The propulsion module was jettisoned after the required velocity was attained. (For the major part of the mission, "spacecraft" and "mission module" will be used interchangeably. In describing the pre-launch configuration and launch phase, "spacecraft" will refer to the combined "mission module" and "propulsion module.")

The launch weight of the spacecraft was 2,066 kilograms (4,556 pounds), including the propulsion module with its large, solid-propellant rocket motor that weighed 1,207 kilograms (2,660 pounds). After injection the mission module weighed 825 kilograms (1,819 pounds) including a 117 kilograms (258 pound) science instrument payload.

Like the Mariners that explored the inner planets and the Viking Mars orbiters, the Voyagers are stabilized on three axes using the Sun and a star as celestial references.

Hot gas jets provide thrust for attitude stabilization and for trajectory correction maneuvers.

Three engineering subsystems are programmable for on-board control of most spacecraft functions. The three are the computer command subsystem (CCS), flight data subsystem (FDS) and attitude and articulation control subsystem (AACS). The memories of the units can be updated or modified by ground command.

A nuclear power source -- three radioisotope thermo-electric generators -- provides electrical power for the space-craft.

The pointable science instruments are mounted on a two-axis scan platform at the end of the science boom. Other body-fixed and boom-mounted instruments are aligned to provide for proper interpretation of their measurements.

Data storage capacity on the spacecraft is about 536 million bits -- approximately the equivalent of 100 full-resolution photos.

Dual-frequency communication links -- S-band and X-band -- provide accurate navigation data and large amounts of science information during planetary encounter periods (up to 115,200 bits per second at Jupiter and 44,800 bps at Saturn).

The dominant feature of the spacecraft is the 3.66-meter (12-foot) diameter high-gain antenna that, after the first 80 days of flight, almost continuously points toward Earth. Periodic exceptions are during trajectory-correction, calibration, or science data acquisition attitude maneuvers.

While the high-gain antenna dish is white, most visible parts of the spacecraft are black -- blanketed or wrapped for thermal control and micrometeoroid protection. A few small areas are finished in gold foil or have polished aluminum surfaces.

Structure and Configuration

The basic mission module structure is a 24.5-kilogram (54-pound) 10-sided aluminum framework with 10 electronics packaging compartments. The structure is 47 centimeters (18.5 inches) high and 178 centimeters (70 inches) across from flat to flat; 188 centimeters (74 inches) from longeron to opposite longeron. The electronics assemblies are structural elements of the 10-sided box.

The spherical propellant tank that contains fuel for hydrazine thrusters for attitude control and trajectory correction maneuvers occupies the center cavity of the decagon. Propellant lines carry hydrazine to 12 small attitude-control and four TCM thrusters on the mission module and to larger thrust-vector-control engines on the propulsion module during launch.

The 3.66-meter (12-foot) diameter high-gain parabolic reflector is supported above the basic structure by a tubular truss work. The antenna reflector has an aluminum honeycomb core and is surfaced on both sides by graphite epoxy laminate skins. Attachment to the trusses is along a 178-centimeter (70-inch) diameter support ring. The Sun sensor protrudes through a cutout in the antenna dish. An X-band feed horn is at the center of the reflector. Two S-band feed horns are mounted back-to-back with the X-band sub-reflector on a three-legged truss above the dish. One radiates through the sub-reflector, transparent at S-band, to the high-gain dish. The other functions as the low-gain antenna.

Louver assemblies for temperature control are fastened to the outer faces of two electronics compartments -- those housing the power conditioning assembly and the radio transmitter power amplifiers. The top and bottom of the 10-sided structure are enclosed with multi-layer thermal blankets.

Two Canopus star tracker units are mounted side-by-side and parallel atop the upper ring of the decagon.

Three radioisotope thermoelectric generators are assembled in tandem on a deployable boom hinged on an outrigger arrangement of struts attached to the basic structure. The RTG boom is constructed of steel and titanium. Each RTG unit, contained in a beryllium outer case, is 40.6 centimeters (16 inches) in diameter, 51 centimeters (20 inches) long and weighs 39 kilograms (86 pounds).

The science boom, supporting the instruments that measure energetic particle radiation, is located 180 degrees from the RTG boom and is hinged to a truss extending out from the decagon behind the high-gain antenna. The boom, 2.3 meters (7 1/2 feet) long, is a bridgework of graphite epoxy tubing. Attached on opposite sides of the boom at its mid-point are the cosmic ray and low-energy charged particle instruments. Farther out on the boom is the plasma science instrument.

The two-axis scan platform is mounted at the end of the boom and provides precision pointing for four remote-sensing instruments -- the ultraviolet spectrometer, infrared interferometer spectrometer amd radiometer, photopolarimeter (no longer operating on Voyager 1) and a two-camera imaging science subsystem. Total platform gimballed weight is 100 kilograms (220 pounds).

With both the RTG and science booms deployed, the nearest boom-mounted instrument to a radiation source is 4.8 meters (16 feet), with the bulk of the spacecraft between the two. The closest platform-mounted instrument is 6.4 meters (21 feet) away.

A pair of 10-meter (33-foot) whip antennas, deployed from a position outside the top ring of the basic structure and extending down between the RTG boom outrigger members, are part of the planetary radio astronomy (PRA) instrument package and are shared with the plasma wave instrument (PWS). The PRA and PWS assemblies are body-mounted adjacently. The beryllium-copper tubing antennas, which form a right angle, were rolled flat in their housing before deployment by small electric motors.

The magnetic fields experiment consists of an electronics subassembly located in one of the mission module electronics bays and four magnetometers -- two high-field sensors affixed to the spacecraft and two low-field sensors mounted on a 13-meter (43-foot) deployable boom. The boom, constructed of epoxy glass, spiralled from its stowed position in an aluminum cylinder to form a rigid triangular mast with one magnetometer attached to its end plate and another positioned 6 meters (20 feet) closer to the spacecraft. The mast weighs 2.26 kilograms (five pounds), a few ounces less than the cabling running its length and carrying power to and data from the magnetometers. The boom housing is a 2.8-centimeter (nine-inch) diameter cylinder, 66 centimeters (26 inches) long, supported by the RTG outrigger. The mast uncoils in helix fashion along a line between the rear face of the high-gain antenna and the RTG boom.

The basic structure of the discarded propulsion module was a 43.54-kilogram (96-pound) aluminum semi-monocoque shell. The cylinder, 99 centimeters (39 inches) in diameter and 89 centimeters (35 inches) long, was suspended below the mission module structure by an eight-member tubular truss adapter. The hollow of the structure contained the solid rocket motor that delivered the final powered stage of flight. The rocket, which weighed 1,123 kilograms (2,475 pounds) including 1,039 kilograms (2,290 pounds) of propellant, developed an average 15,300 pounds thrust during its 43-second burn duration.

Mounted on outriggers from the structure were eight hydrazine engines that provided attitude control during the solid-motor burn. Hydrazine fuel was supplied from the mission module.

A pair of batteries and a remote driver module for powering the valve drivers to the thrust-vector-control engines were positioned on the outer face of the cylindrical propulsion module structure.

A four-square-foot shunt radiator/science calibration target faces outward from the propulsion module truss adapter toward the scan platform. The dual-purpose structure is a flat sandwich of two aluminum radiating surfaces lining a honeycomb core. Through power collectors and emitter resistors between the plates, any amount of the electrical power from the RTGs can be radiated to space as heat. The outer surface also serves as a photometric calibration target for the remote-sensing science instruments on the scan platform.

The shunt radiator and the propulsion-module truss adapter remained when the propulsion module was jettisoned.

Steel alloy/gold foil plume deflectors extended from the propulsion module to shield the stowed RTGs and scan platform from rocket exhaust during engine firing.

The spacecraft adapter, a truncated aluminum cone, joined the propulsion module to the Centaur stage of the launch vehicle. The adapter, 76 centimeters (30 inches) tall, was 160 centimeters (63 inches) in diameter at the base (Centaur attachment), 99 centimeters (39 inches) at the spacecraft separation joint and weighed 36 kilograms (79 pounds). The adapter remained with the Centaur rocket stage at spacecraft separation.

Launch Configuration

Some mechanical elements of the spacecraft were rigidly restrained against the severe vibration during launch. After launch, appendages that had been latched securely within the Centaur stage nose fairing were deployed to their cruise positions.

The pyrotechnic subsystem provided simple and positive deployment with explosive squibs. Devices stowed securely during launch and released for deployment by the pyrotechnic subsystem were the science boom, RTG boom and magnetometer boom. Uncertainty concerning the full deployment and locking of the science boom on Voyager 2, first spacecraft launched, existed for several weeks.

The pyrotechnic subsystem also routed power to devices to separate spacecraft from launch vehicle, activated the propulsion module batteries, ignited the solid-propellant rocket motor, sealed off the propellant line carrying hydrazine from the mission module to the propulsion module, jettisoned the propulsion module and released the IRIS dust cover. That subsystem is no longer used.

Communications

Communication with the Voyagers is by radio link between Earth tracking stations and a dual-frequency radio system aboard each spacecraft.

The uplink operates at S-band only, carrying commands and ranging signals from ground stations to one of a pair of redundant receivers. The downlink is transmitted from the spacecraft at both S-band and X-band frequencies.

The on-board communications system also includes a programmable flight data subsystem (FDS), modulation/demodulation subsystem (MDS), data storage subsystem (DSS) and high-gain and low-gain antennas.

The FDS, one of the three on-board computers, controls the science instruments and formats all science and engineering data for telemetry to Earth. The telemetry modulation unit (TMU) of the MDS feeds data to the downlink. The flight command unit (FCU) of the MDS routes ground commands received by the spacecraft.

Only one receiver is powered at any one time, with the redundant receiver at standby. The receiver operates continuously during the mission at about 2113 megahertz. Different frequency ranges have been assigned to the radio-frequency subsystem of each spacecraft. The receiver can be used with either the high-gain or low-gain antenna. Voyager 2's primary receiver failed on April 5, 1978, and the spacecraft is operating on its backup receiver, which is also crippled with a partial failure of that portion of the receiver that acquires and tracks the transmitted signal.

(Appendix A contains a detailed description of Voyager 2 communication problem, the process used to accommodate the operations limitation imposed by the problem, and some of the communications features of the Voyager 2 Saturn encounter.)

The S-band transmitter consists of two redundant exciters and two redundant RF power amplifiers, of which any combination is possible. Only one exciter-amplifier combination operates at any one time. Selection of the combination is by on-board failure-detection logic within the computer command subsystem (CCS), with ground-command backup. The same arrangement of exciter-amplifier combinations makes up the X-band transmitting unit.

One S-band and both X-band amplifiers employ traveling wave tubes (TWT). The second S-band unit is a solid state amplifier. The S-band transmitter is capable of operating at 9.4 watts or at 28.3 watts when switched to high power and can radiate from both antennae. X-band power output is 12 watts and 21.3 watts. X-band uses only the high-gain antenna. (S-band and X-band never operate at high power simultaneously.)

When no uplink signal is being received, the transmitted S-band frequency of about 2295 MHz and X-band frequency of 8418 MHz originate in the S-band exciter's auxiliary oscillator or in a separate ultra-stable oscillator (one-way tracking). With the receiver phase-locked to an uplink signal, the receiver provides the frequency source for both transmitters (two-way tracking). The radio system can also operate with the receiver locked to an uplink signal while the downlink carrier frequencies are determined by the on-board oscillators (two-way noncoherent tracking).

Both the 64-meter (210-foot) and 34-meter (112-foot) antenna stations of the Deep Space Network can receive the downlink X-band signal. The entire 64-meter, 34-meter and 26-meter (85-foot) antenna stations are capable of receiving at S-band.

The X-band downlink was not normally used during the first 80 days of the mission -- until Earth was within the beam of the spacecraft's high-gain antenna. Communications during launch, near-Earth and early cruise phase operations were confined to S-band and the low-gain antenna. An exception occurred early in the flight when the spacecraft, on inertial control, pointed the high-gain antenna toward Earth to support instrument calibration and an optical navigation/high-rate telecommunications link test. During its calibration sequence on Sept. 18, 1977, Voyager 1 took pictures of Earth-Moon system.

The high-gain antenna, with 3.66-meter-diameter (12 feet) parabolic reflector, provides a highly directional beam. The low-gain antenna provides essentially uniform coverage in the direction of Earth.

Under normal conditions, after the first 80 days of the mission, all communications -- both S-band and X-band -- have been via the high-gain antenna. The normal communication configuration for the Voyager 2 Saturn encounter is for the X-band transmitter to operate at high power and the S-band transmitter to operate at low power. The only variations to that configuration are during Earth-occultation entrance and exit when the S-band is at high power and the X-band is at low -power. Low-power X-band is also used during two IRIS calibrations, when both the S-band and X-band transmitters are at low power.

Commanding the Spacecraft

Ground commands are used to put into execution selected flight sequences, to cause execution of discrete events, or to cope with unexpected events. Commands are issued in either a pre-determined, timed sequence via on-board program control or directly as received from the ground. Most commands are issued by the spacecraft's computer command subsystem (CCS) in its role as "sequencer-of-events" and by the flight data subsystem (FDS) as controller of the science instruments.

All communications between spacecraft and Earth are in digital form. Command signals, transmitted at 16 bits per second (bps) to the spacecraft, are detected in the flight command unit and routed to the CCS for further routing to their proper destination. Ground commands to the spacecraft fall into two major categories: discrete commands (DC), and coded commands (CC).

A discrete command causes a single action on the spacecraft. For example, DC-2D switches the S-band amplifier to high power; DC-2DR, S-band amplifier low power; DC-2E, S-band transmits from high-gain antenna; DC-2ER, S-band transmits low-gain. Coded commands are the transfer of digital data from the computer command system or from the ground via the CCS to user subsystems. Subsystems receiving coded commands are flight data, attitude and articulation control, modulation/demodulation, data storage and power.

Downlink Telemetry

Data telemetered from the spacecraft consists of engineering and science measurements prepared for transmission by the flight data subsystem, telemetry modulation unit and data storage subsystem. The encoded information will indicate voltages, pressures, temperatures, television pictures and other values measured by the spacecraft telemetry sensors and science instruments.

Two telemetry channels -- low rate and high rate -- are provided for the transmission of spacecraft data. The low rate channel functions only at S-band at a single 40-bits-per-second data rate and contains real-time engineering data exclusively. It is on only during planetary encounters when the high-rate channel is operating at X-band.

The high-rate channel is on throughout the mission. It operates at either S-band or X-band and contains the following types of data:

- Engineering only at 40 bits per second or 1,200 bps (the higher data rate usually occurs only during launch and trajectory correction maneuvers) transmitted at S-band only.
- Real-time cruise science and engineering at 2,560, 1,280, 640, 320, 160 and 80 bps. Reduced data rates result in reduced instrument sampling. Data rate is reduced only when the telecommunications link cannot support the higher rate.
- Real-time encounter general science and engineering at 7.2 kilobits per second (a special 115.2 kbps rate was available at Jupiter for the planetary radio astronomy and plasma wave instruments) transmitted at X-band only.
- Real-time encounter general science, engineering and

television at 115.2, 89.6, 67.2, 44.8, 29.866 2/3 and 19.2 kbps transmitted at X-band only. The maximum usable data rate at Saturn is 44.8 kbps.

- Real-time encounter general science and engineering, plus tape recorder playback, at 67.2 and 44.8 kbps and 29,866 2/3 bps transmitted at X-band only. The maximum usable data rate at Saturn is 44.8 kbps.
- Play back recorded data only at 21.6 and 7.2 kbps transmitted at X-band only.
- Memory data stored in the three on-board computers -- CCS, FDS and AACS -- read out and played back at 40 or 1200 bps transmited at either S-band or X-band (treated as engineering data).

The many data rates for each type of telemetered information are required by the changing length of the telecommunications link with Earth and the possible adverse effects of Earth weather upon reception of X-band radio signals.

In order to allow real-time transmission of video information at each encounter, the flight data subsystem can handle the imaging data at six downlink rates from 115.2 to 19.2 kbps. The 115.2-kbps rate represents the standard full-frame readout (at 48 seconds per frame) of the TV vidicon. Under normal conditions, that rate was used at Jupiter. Full-frame, full-resolution TV from Saturn can be obtained by increasing the frame readout time to 144 seconds (3:1 slow scan) and transmitting the data at 44.8 kbps. A number of other slow scan and frame-edit options are available to match the capability of the telecommunications link.

The data-storage subsystem is a digital tape recorder which can record at two rates: TV pictures, general science and engineering at 115.2 kbps; general science and engineering at 7.2 kbps; and engineering only at 7.2 kbps (engineering is acquired at only 1,200 bps, but is formatted with filler to match the recorder input rate). The tape transport is belt-driven. Its 1/2-inch magnetic tape is 328 meters (1,075 feet) long and is divided into eight tracks that are recorded sequentially one track at a time. Total recycleable storage capacity is about 536 million bits -- the equivalent of 100 TV pictures. Playback is at one of four speeds -- 57.6, 33.6, 21.6 and 7.2 kbps.

Downlink telemetry data rates are maximized during the Saturn encounter by the simultaneous receipt and processing of the received signal from the spacecraft by two co-located ground antennas (one 64-meter and one 34-meter). Goldstone, Spain and Australia have co-located stations providing the arrayed capability. Combining the received spacecraft signal results in approximately a one-1dB improvement in signal-to-noise ratio. That is equivalent to having one 72-meter station at each location. Loss of a 34-meter station during any arrayed track would result in a reduction of data quantity or quality, but not in the loss of all data. Loss of the 64-meter antenna would result in the loss of X-band capability, reducing available data to engineering information only.

Tracking the Spacecraft for Navigation

Very precise navigation is required to achieve the desired maneuver and flyby accuracies for a multi-planet/satellite encounter mission.

Radiometric observations made by a tracking station on Earth using radio signals to and from the spacecraft are important in determining the spacecraft's position and velocity. The radio data combined with photographs of selected satellites against the background of known stars taken by the cameras provide the information necessary for the precise navigation required. (Appendix B contains a description of the Voyager navigation process, including descriptions of the types of data used, orbit determination, trajectory correction, the Voyager 2 targeting strategy and the Voyager 2 navigation status as of May 15, 1981.)

Power

The Voyager power subsystem supplies all electrical power to the spacecraft by generating, converting, conditioning and switching the power.

Power source for the mission module is an array of three radioisotope thermoelectric generators (RTGs). The propulsion module, active only during the brief injection phase of the mission, used a separate battery source; no other batteries are used on the spacecraft because of the necessity for long mission life.

The RTG units, mounted in tandem on a deployable boom and connected in parallel, convert to electricity the heat released by the isotopic decay of Plutonium-238. Each isotope heat source has a capacity of 2,400 watts thermal with a resultant maximum electrical power output of 160 watts at the beginning of the mission. There is a gradual decrease in power output. The total power available from the three RTGs on each Voyager ranges from about 475 watts within a few hours after launch to 430 watts after the spacecraft pass Saturn.

Spacecraft power requirements from launch to post-Saturn operations are characterized by this general power timeline: launch and post-launch, 235 to 265 watts; interplanetary cruise, 320 to 365 watts; Jupiter encounter, 350 to 370 watts; Saturn encounter, 377 to 382 watts; and post-Saturn, less than 365 watts.

Telemetry measurements have been selected to provide the necessary information for power management by ground command, if needed.

Power from the RTGs is held at a constant 30 volts dc by a shunt regulator. The power is supplied directly to some spacecraft users and is switched to others in the power distribution subassembly. The main power inverter is also supplied 30 volts dc for conversion to 2.4 kilohertz square wave used by most spacecraft subsystems. Again, the ac power may be supplied directly to users or can be switched on or off by power relays.

Command-actuated relays control the distribution of power in the spacecraft. Some relays function as simple on-off switches and others transfer power from one module to another within a subsystem.

Among the users of dc power, in addition to the inverter, are the radio subsystem, gyros, propulsion isolation valves, some science instruments, most temperature control heaters and the motors that deloyed the planetary radio astronomy antennas. Other elements of the spacecraft use the 2.4 khz power.

There are two identical 2.4 khz power inverters -- main and standby. The main inverter is on from launch and remains on throughout the mission. In case of a malfunction or failure in the main inverter, the power chain, after a 1.5-second delay, is switched automatically to the standby inverter. Once the switch-over is made, it is irreversible.

A 4.8 khz sync and timing signal from the flight data subsystem is used as a frequency reference in the inverter. The frequency is divided by two and the output is 2.4 khz plus-or-minus 0.002%. This timing signal is sent, in turn, to the computer command subsystem, which contains the spacecraft's master clock.

Because of the long mission lifetime, charged capacitor energy-storage banks are used instead of batteries to supply the short-term extra power demanded by instantaneous overloads that would cause the main dc power voltage to dip below acceptable limits. A typical heavy transient overload occurs at turn-on of a radio power amplifier.

Full output of the RTGs, a constant power source, must be used or dissipated in some way to prevent overheating of the generator units and dc voltage rising above allowed maximum. Full use is controlled by a shunt regulator that dumps excess RTG output power above that required to operate the spacecraft. The excess power is dissipated in resistors in the shunt radiator mounted outside the spacecraft and radiated into space as heat.

Two batteries independently supplied unregulated dc power to a remote driver module (RDM) for powering valve drivers to the thrust-vector-control engines on the propulsion module during the injection phase of the mission. The batteries and the RDM are located in the propulsion module that was jettisoned a few minutes after the mission module was placed on the Jupiter trajectory. Each battery was composed of 22 silver oxide-zinc cells with a capacity of 1,200 ampere seconds at 28 to 40 volts, depending upon the load.

Basic requirement on the batteries was high power for a short period -- 12 minutes. With a lifetime of only 66 minutes, the batteries were kept inert until just four seconds before Centaur separation and 20 seconds before solid rocket ignition. After activation, in which an electrolyte was injected into the cells, the batteries were at full voltage in one-half second and ready for use in two seconds.

Computer Command Subsystem

The heart of the on-board control system is the computer command subsystem (CCS), which provides a semi-automatic capability to the spacecraft.

The CCS includes two independent memories, each with a capacity of 4,096 data words. Half of each memory stores reusable fixed routines that do not change during the mission. The second half is reprogrammable by updates from the ground.

Most commands to other spacecraft subsystems are issued from the CCS memory, which, at any given time, is loaded with the sequence appropriate to the mission phase. The CCS also can decode commands from the ground and pass them along to other spacecraft subsystems.

Under control of an accurate on-board clock, the CCS counts hours, minutes or seconds until some pre-programmed interval has elapsed and then branches into subroutines stored in the memory that results in commands to other subsystems. A sequencing event can be a single command or a routine that includes many commands (e.g., manipulating the tape recorder during a playback sequence).

The CCS can issue commands singly from one of its two processors or in parallel or tandem from both processors. An example of CCS dual control is the execution of trajectory correction maneuvers.

TCM thrusters are started with a tandem command (both processors must send consistent commands to a single output unit) and stopped with a parallel command (either processor working through different output units will stop the burn).

The CCS can survive any single internal fault; each functional unit has a duplicate elsewhere in the subsystem.

Attitude Control and Propulsion

Stabilization and orientation of the spacecraft from launch-vehicle separation until end of the mission is provided by two major engineering subsystems -- attitude and articulation control (AACS) and propulsion.

Propulsion Subsystem

The propulsion subsystem consisted of a large solid-propellant rocket motor for final Jupiter trajectory velocity and a hydrazine blowdown system that fuels 16 thrusters on the mission module and eight reaction engines on the propulsion module. (The propulsion module was jettisoned after injection.)

The single hydrazine (N_2H_4) supply is contained within a 28-inch-diameter spherical titanium tank, separated from the helium pressurization gas by a Teflon-filled rubber bladder. The tank, located in the central cavity of the mission module's 10-sided basic structure, contained 104 kilograms (230 pounds) of hydrazine at launch and was pressurized at 420 psi. As the propellant is consumed, the helium pressure will decrease to a minimum of about 130 psi.

All 24 hydrazine thrusters use a catalyst that spontaneously initiates and sustains rapid decomposition of the hydrazine.

The 16 thrusters on the mission module each deliver 0.2-pound-thrust. Four are used to execute trajectory correction maneuvers; the others in two redundant six-thruster branches, to stabilize the spacecraft on its three axes. Only one branch of attitude control thrusters is needed at any time.

Mounted on outriggers from the propulsion module are four 100-pound-thrust engines that, during solid-motor burn, provided thrust-vector control on the pitch and yaw axes. Four five-pound-thrust engines provided roll control.

The solid rocket, which weighed 1,123 kilograms (2,475 pounds) including 1,039 kilograms (2,290 pounds) of propellant, developed an average 15,300 pounds thrust during its 43-second burn duration.

Attitude and Articulation Control Subsystem

The AACS includes an onboard computer called HYPACE (hybrid programmable attitude control electronics), redundant sun sensors, redundant Canopus star trackers, three two-axis gyros and scan actuators for positioning the science platform.

The HYPACE contains two redundant 4,096-word plated-wire memories -- part of which are fixed and part reprogrammable -- redundant processors and input/output driver circuits.

Injection Propulsion Control

Because of the energy required to achieve a Jupiter ballastic trajectory with an 825-kilogram (1,819-pound) payload, the spacecraft launched by the Titan III E/Centaur included a final propulsive stage that added a velocity increment of about two kilometers per second (4,475 miles per hour).

The solid-rocket motor in the propulsion module was ignited 15 seconds after the spacecraft separated from the Centaur and burned for about 43 seconds. Firing circuits to the motor were armed during launch-vehicle countdown.

The four 100-pound-thrust engines provided pitch and yaw attitude control and the four five-pound-thrust engines provided roll control until burnout of the solid rocket motor. The hydrazine engines responded to instructions from AACS's computer. Only two pitch/yaw and two roll engines at most were thrusting at any given time.

Before solid rocket ignition and after burnout, until the propulsion module separated from the mission module, only the five-pound-thrust roll engines were required for attitude control They were oriented on the propulsion module so that, by proper engine selection by the AACS, attitude control was maintained on all three axes.

Approximately 11 minutes after solid-rocket burnout, the propulsion module was jettisoned. Several seconds earlier, the propellant line carrying hydrazine from the mission module to the propulsion module was sealed and separated.

Celestial Reference Control

The sun sensors, which look through a slot in the high-gain antenna dish, are electro-optical devices that send attitude position error signals to HYPACE, which, in turn, signals the appropriate attitude control thrusters to fire and turn the spacecraft in the proper direction. Sun lock stabilizes the spacecraft on two axes (pitch and yaw).

The star Canopus, one of the brightest in the galaxy, is usually the second celestial reference for three-axis stabilization. Two Canopus trackers are mounted so that their lines of sight are parallel. Only one is in use at any one time. The star tracker, through HYPACE logic, causes the thrusters to roll the spacecraft about the already-fixed Earth or Sun-pointed roll axis until the tracker is locked on Canopus. Brightness of the tracker's target star is telemetered to the ground to verify the correct star has been acquired.

To enhance downlink communication capability, the Sun sensor is electrically biased (offset) by commands from the computer command subsystem to point the roll axis at or as near the Earth as possible. The sun sensor can be biased plus and minus 20° in both pitch and yaw axes.

Three-axis stabilization with celestial reference is the normal attitude-control mode for cruise phases between planets.

Inertial Reference Control

The spacecraft can be stabilized on one axis (roll) or all three axes with the AACS's inertial reference unit consisting of three gyros.

Appropriate inertial reference modes are used whenever the spacecraft is not on sun/star celestial lock. Such situations include maintaining inertial reference from Centaur separation until initial celestial acquisition is achieved; purposely turning the spacecraft off sun/star lock to do directed trajectory corrections or science instrument observations or calibrations; providing a reference when the sun is occulted; and providing a reference when concern exists that the Canopus or Sun sensor will detect stray intensity from unwanted sources -- planets, rings, satellites.

Each gyro has associated electronics to provide position information about two orthogonal axes (Gyro A: pitch and yaw, Gyro B: roll and pitch, Gyro C: yaw and roll). Normally, two gyros are on for any inertial mode. The gyros have two selectable rates, one -- high rate -- for propulsion-module injection phase; the other for mission-module cruise and trajectory-correction and science maneuvers.

Trajectory Correction Maneuvers

The Voyager trajectories are planned around eight trajectory correction maneuvers (TCM) with each spacecraft between launch and Saturn encounter. Mission requirements call for extremely accurate maneuvers to reach the desired zones at Jupiter, Saturn and the target satellites. Total velocity increment capability for each spacecraft is about 190 meters per second (425 miles per hour).

TCM sequencing is under control of the computer command subsystem (CCS), which sends the required turn angles to the AACS for positioning the spacecraft at the correct orientation in space and, at the proper time, sends commands to the AACS to start and stop the TCM burn. Attitude control is maintained by pulse-off sequencing of the TCM engines and pulse-on sequencing of two attitude-control roll thrusters. Position and rate signals are obtained from the gyros. After the burn, reacquisition of the cruise celestial references is accomplished by a second set of spacecraft turns. All TCMs are enabled by ground command.

Science Platform (Articulation Control)

Voyager's two television cameras, ultraviolet spectrometer, photopolarimeter and infrared spectrometer and radiometer are mounted on the scan platform that can be rotated about two axes for precise pointing at Jupiter, Saturn and their moons during the planetary phases of the flight. The platform is located at the end of the science boom. Total gimballed weight is 100 kilograms (220 pounds).

Controlled by the attitude and articulation control subsystem (AACS), the platform allows multiple pointing directions of the instruments. Driver circuits for the scan actuators -- one for each axis -- are located in the AACS computer. The platform's two axes of rotation are described as the azimuth angle motion about an axis displaced 7° from the spacecraft roll axis (perpendicular to the boom centerline) and elevation angle motion about an axis perpendicular to the azimuth axis and rotating with the azimuth axis. Angular range is 360° in azimuth and 210° in elevation.

The platform is slewed one axis at a time with selectable slew rates in response to computer command subsystem commands to the AACS. Slew rates are: 1.0°/s; o.33°/s; .083°/s; and a special

UVS low rate: .0052°/s. Camera line of sight is controlled to within 2.5 milliradians per axis.

Temperature Control

The two Voyager spacecraft are designed to operate farther from Earth than any previous man-made object. Survival depends greatly upon keeping temperatures within operating limits while the spacecraft traverses an environmental range of one solar constant at the Earth distance to four percent of that solar intensity at Jupiter and one per cent at Saturn.

Unprotected surfaces at the Saturn range, nearly one billion miles from the Sun, can reach 320°F below zero -- the temperature of liquid nitrogen.

Both top and bottom of the mission module's basic decagon structure are enclosed with multi-layer thermal blankets to prevent the rapid loss of heat to space. The blankets are sandwiches of aluminized Mylar, sheets of Tedlar for micrometeroid protection and outer black Kapton covers that are electrically conductive to prevent the accumulation of electrostatic charges.

Also extensively blanketed are the instruments on the scan platform. Smaller blankets and thermal wrap cover eight electronics bays, boom and body-mounted instruments, cabling, propellant lines and structural struts. Only a few exterior elements of the spacecraft are not clad in the black film -- the high-gain antenna reflector, plasma sensors, sun sensors and antenna feed cones.

Temperature control of four of the 10 electronics compartments is provided by thermostatically controlled louver assemblies that provide an internal operating range near room temperature. The louvers are rotated open by bimetallic springs when large amounts of heat are dissipated. These bays contain the power-conditioning equipment and the radio power amplifiers. Mini-louvers are located on the scan platform, cosmic ray instrument and the sun sensors.

Radioisotope heating units (RHU), small non-power-using heat elements that generate one watt of thermal energy, are located on the magnetometer sensors and the sun sensors. No RHUs are used near instruments that detect charged particles. Electric heaters are located throughout the spacecraft to provide additional heat during portions of the mission. Many of the heaters are turned off when their respective valves, instruments or subassemblies are on and dissipating power.

VOYAGER EXPERIMENTS

COSMIC-RAY EXPERIMENT

The cosmic-ray experiment has four principal scientific objectives:

1. To measure the energy spectrum of electrons and cosmic-ray nuclei.
2. To determine the elemental and isotopic composition of cosmic-ray nuclei.
3. To make elemental and isotopic studies of Jupiter's radiation belts and to explore Saturn's environment.
4. To determine intensity and directional characteristics of energetic particles as a function of radial distance from the Sun, and determine location of the modulation boundary.

The cosmic-ray experiment uses multiple solid-state detector telescopes to provide large solid-angle viewing; the low-energy-telescope system covers the range from 0.5 to 9 million electron volts (MeV) per nucleon; the high-energy telescope system covers the range from 4 to 500 million electron volts. The electron telescope system covers the range from 7 million to 100 million electron volts.

The cosmic-ray experiment weighs 7.52 kilograms (16.57 pounds) and uses 5.2 watts of power.

Dr. Rochus E. Vogt of the California Institute of Technology is principal investigator.

LOW-ENERGY CHARGED PARTICLE EXPERIMENT

Scientific objectives of the Low-Energy Charged-Particle Experiment include studies of the charged-particle composition, energy-distribution and angular distribution with respect to:

1. Saturn's magnetosphere (exploratory) and Jupiter's magnetosphere (detailed studies).
2. Interactions of charged particles with the satellites of Jupiter and Saturn and possibly with the rings of Saturn.
3. Measurements of quasi-steady interplanetary flux and high-energy components of the solar wind.
4. Determination of the origin and interstellar propagation of galactic cosmic rays (those that come from outside the solar system).
5. Measurements of the propagation of solar particles in the outer solar system.

The experiment uses two solid-state detector systems on a rotating platform mounted on the scan platform boom. One system is a low-energy magnetospheric particle analyzer with large dynamic range to measure electrons with energy ranging from 15,000 electron volts (15 KeV) to greater than 1 million electron volts (1 MeV); and ions in the energy range from 15,000 electron volts per nucleon to 160 million electron volts per nucleon.

The second detector system is a low-energy particle telescope that covers the range from 0.15 million electron volts per nucleon to greater than about 10 million electron volts per nucleon.

The Low-Energy Charged-Particle Experiment weighs 7.47 kilograms (16.47 pounds) and draws 3.9 watts during cruise and 4.2 watts during planetary encounter.

Dr. S. M. (Tom) Krimigis of the Applied Physics Laboratory, Johns Hopkins University, is principal investigator.

MAGNETIC FIELDS EXPERIMENT

The magnetic field of a planet is an externally measurable manifestation of conditions deep in its interior.

The magnetic-fields instruments on Voyager 1 and 2 determine the magnetic field and magnetospheric structure at Jupiter and Saturn; they study the interaction of the magnetic field and the satellites that orbit the planets inside that field and study the interplanetary-interstellar magnetic field.

Four magnetometers are carried aboard Voyager. Two are low-field, three-axis instruments located on a boom to place them as far from the spacecraft body as possible. That allows separation of the spacecraft's magnetic field from the external field that is to be measured. The other two magnetometers are high-field, three-axis instruments mounted on the spacecraft body.

The boom-mounted, low-field instruments measure the magnetic fields in the range from 0.002 gamma to 50,000 gamma. (Fifty-thousand gamma equals one-half gauss, about the average magnetic field strength at the surface of Earth.)

The high-field instruments cover the range from 12 gamma to 20 gauss. While the highest field strengths measured by the

Pioneer spacecraft at Jupiter were about 14 gauss, scientists expect that localized, stronger fields may be associated with the planets or some of their satellites.

Total weight of the magnetic-fields experiment is 5.5 kilograms (12 pounds). The experiment uses 3.2 watts of power.

Dr. Norman Ness of NASA's Goddard Space Flight Center is principal investigator.

INFRARED INTERFEROMETER SPECTROMETER AND RADIOMETER

The IRIS is designed to perform spectral and radiometric measurements of the Jovian, Saturnian and Uranian planetary systems, and targets of opportunity during the cruise phases.

Scientific objectives for IRIS are:

1. Measurement of the energy balance of Jupiter, Saturn and Uranus.
2. Studies of the atmospheric compositions of Jupiter, Saturn, Titan and other satellites, and Uranus.
3. Temperature, structure and dynamics of the atmospheres.
4. Measurements of composition and characteristics of clouds and aerosols.
5. Studies of the composition and characteristics of ring particles (at Saturn) and the surfaces of those satellites the instrument will observe.

The instrument provides broad spectral coverage, high spectral resolution and low noise-equivalence-radiance through use of dual interferometers. That and the variable resolution of the instrument, as well as the precision of the radiometer, will allow scientists to acquire information about a wide variety of scientific questions concerning the atmospheres of the planets and satellites, local and global energy balance, and the nature of satellite surfaces and the rings of Saturn.

The instrument has two fields of view from its position on the scan platform. The first is centered on the boresight of the 51-centimeter (20-inch) Cassegrain telescope. The second field of view -- for solar calibration -- is pointed 20 degrees off the telescope boresight. It approximately overlaps the ultraviolet spectrometer's occultation field of view. IRIS pointing is controlled by the scan platform.

Dr. Rudolf A. Hanel of Goddard Space Flight Center is principal investigator.

PHOTOPOLARIMETER

A great deal of information about the composition of an object can be learned from the way that object reflects light. The light that Voyager's photopolarimeter measures is polarized by the chemicals and aerosols in the atmospheres and small particles on the surfaces.

The photopolarimeter studies aerosol particles in the atmospheres of the planets and satellites, and the textures and compositions of the surfaces of satellites. It also will measure size, albedo, spatial distribution, shape and orientation of particles in Saturn's rings; and optical and geometric thickness of the rings.

The instrument is made up of a 15-centimeter (6-inch) Cassegrain telescope, aperture sector, polarization analyzer wheel, filter wheel and a photomultiplier-tube detector. The filter wheel carries eight filters ranging from 2,350-Angstrom to 7,500-Angstrom wavelength; five linear polarizers (0, 60, 120, 135 and 45 degrees) plus an open position (no polarizer) and a dark slide. The instrument's field of view can be set at 3.5 degrees, 1 degree, 1/4 degree and 1/10 degree.

During the Jupiter flyby, Voyager 2's photopolarimeter suffered some radiation damage that affected some of its central logic. As a consequence some of its color filters and polarizers cannot be used. The Saturn encounter will concentrate on the use of deep ultraviolet (2,800 Angstroms) and deep red (7,500 Angstroms) and perpendicular polarizations (45 and 135 degrees). Those choices will return much of the scientific data the photopolarimeter team had counted on.

The photopolarimeter weighs 4.41 kilograms (9.7 pounds) and uses 2.4 watts average power.

Dr. Arthur L. Lane of the Jet Propulsion Laboratory is principal investigator.

PLANETARY RADIO ASTRONOMY

The Planetary Radio Astronomy experiment consists of a stepped frequency radio receiver that covers the range from 20 kilohercz to 40.5 megahertz, and two monopole antennas 10 meters (33 feet) long, to detect and study a variety of radio signals emitted by Jupiter and Saturn.

Scientific objectives of the experiment include detection and study of radio emissions from Jupiter and Saturn and their sources and relationship to the satellites, the planets' magnetic fields, atmospheric lightning and plasma resonance. The instrument also measures planetary and solar radio bursts from new directions in space and relates them to measurements made from Earth.

The receiver is designed to provide coverage in two frequency bands -- one covering the range from 20.4 kilohertz to 1,345 kilohertz, the second from 1,228.8 kilohertz to 40.5 megahertz. The receiver bandwidth is 1 kilohertz in the low-frequency range and 200 kilohertz in the high-frequency band. There are three signal input attenuators to provide switchable total attenuation from 0 to 90 decibels.

The instrument weighs 7.66 kilograms (16.8 pounds) and draws 6.8 watts of power.

Principal investigator is Dr. James W. Warwick of Radiophysics, Inc., Boulder, Colo.

PLASMA EXPERIMENT

Plasma, clouds of ionized gases, moves through the interplanetary region and comes from the Sun and from stars. The plasma experiment uses two Faraday cup plasma detectors, one pointed along the Earth-spacecraft line, the other at right angles to that line.

Scientific objectives of the plasma experiment are:

1. Determine properties of the solar wind, including changes in the properties with increasing distance from the Sun.
2. Study of the magnetospheres that are intrinsic to the planets themselves and that co-rotate with the planets independent of solar wind activity.
3. Study of the satellites of Jupiter and Saturn and the plasma environment of Io.
4. Detection and measurement of interstellar ions.

The Earth-pointing detector uses a novel geometrical arrangement that makes it equivalent to three Faraday cups and determines microscopic properties of the plasma ions. With this detector, accurate values of the velocity, density and pressure can be determined for plasma from the Earth (1 A.U.) to beyond Saturn (10 A.U.). Two sequential energy scans are employed to allow the instrument to cover a broad range of energies -- from 10 electron volts (eV) to 6,000 electron volts (6 KeV). Significant measurements can be made between subsonic and supersonic speeds in cold solar wind or hot planetary magnetosheath.

The variable energy resolution allows scientists to detect and sort out ions that flow with the solar wind at the same time they are measuring the solar wind's properties.

The instrument has a large (180-degree) field of view at planetary encounters and a 90-degree field of view in the solar wind; no electrical or mechanical scanning is necessary.

The other Faraday cup, a side-looking or lateral detector, measures electrons in the range of 10 electron volts to 6 KeV and should improve spatial coverage for any drifting or co-rotating positive ions during planetary encounters.

The instrument was designed primarily for exploring planets' magnetospheres. It is capable of detecting hot subsonic plasma such as has been observed in the Earth's magnetosphere and is expected from ions originating in the McDonough-Brice ring of Io. The instrument's large angular acceptance allows detection of plasma flows well away from the direction of the Sun, such as plasma flows that co-rotate with the planet.

The plasma experiment weighs 9.89 kilograms (21.8 pounds) and draws 8.3 watts of power.

Dr. Herbert Bridge of the Massachusetts Institute of Technology is principal investigator.

PLASMA WAVE

Scientific objectives of the plasma wave experiment are measurements of thermal plasma density profiles at Jupiter and Saturn; studies of wave-particle interaction, and study of the interactions of the Jovian and Saturnian satellites with their planets' magnetospheres.

The plasma-wave instrument measures electric-field components of local plasma waves over the frequency range from 10 hertz to 56 kilohertz.

The experiment shares the two extendable 10-meter (33-foot) electric antennas provided by the planetary radio astronomy experiment team. The two groups use the antennas in different ways. The plasma wave experiment uses the antennas to form a Vee-type balanced electric dipole (while the radio astronomy experiment uses them as a pair of orthogonal monopoles).

In the normal format, the plasma-wave signals are processed with a simple 16-channel spectrum analyzer. At planetary encounters the system provides a spectral scan every four seconds.

The plasma-wave system has a broadband amplifier that uses the Voyager video telemetry link to give electric field waveforms with a frequency range from 50 hertz to 10 kilohertz at selected times during planet encounters.

The experiment is designed to provide key information on the wave-particle interaction phenomena that control important aspects of the dynamics of the magnetospheres of Jupiter and Saturn. Wave-particle interactions play extremely important roles at Earth, and scientists understand that at least the inner magnetosphere of Jupiter is conceptually similar to that of Earth despite the vast differences in size and in the energy of trapped particles.

In addition, the satellites of Jupiter and Saturn appear to provide important localized sources of plasma and field-aligned currents, and they should significantly affect the trapped-particle populations.

The instrument weighs 1.37 kilograms (3.02 pounds) and draws 1.4 watts of power in the step frequency mode and 18 watts in the step frequency plus waveform analyzer mode.

Dr. Frederick L. Scarf of TRW Defense and Space Systems Group is principal investigator.

RADIO SCIENCE

The spacecraft's communications system is used to conduct several experiments by observing how the radio signals are changed on their way to Earth.

By measuring the way signals die out and return when the spacecraft disappears behind a planet or satellite and then reappears, the radio science team can determine the properties of planetary and satellite atmospheres and ionospheres.

The radio signals also allow scientists to make precise measurements of the spacecraft's trajectory as it passes near a planet or satellite. Post-flight analyses allow determination of the mass of a body and its density and shape.

The rings of Saturn will also be explored by the radio science team by measuring the scattering of the radio signals as they travel through the rings. This will provide measurement of ring mass, particle size distribution and ring structure.

The experiment uses the microwave receivers and transmitters on the spacecraft as well as special equipment at the Deep Space Network tracking stations. The spacecraft transmitters are capable of sending 10 or 25 watts at S-band; and 12 or 20 watts at X-band. The spacecraft antenna is a 3.66 meter (12-foot) parabola and is aimed by special maneuvers performed during planet occultations.

Dr. G. L. Tyler of the Center for Radar Astronomy, Stanford University, is team leader.

TELEVISION

The Voyager imaging system is based on those flown aboard Mariner spacecraft, with advancements and changes dictated by the specific requirements of flybys of Jupiter, Saturn and their satellites.

Science objectives for the television experiments include reconaissance of the Jupiter and Saturn systems, including high-resolution photography of atmospheric motions, colors and unusual features (such as the Great Red Spot and similar smaller spots); vertical structure of the atmospheres of the planets; comparative and detailed geology of satellites; satellite size and rotation, and detailed studies of the rings of Saturn.

Two television-type cameras are mounted on the spacecraft's scan platform: a 200-mm focal-length, wide-angle camera with 4,000-Angstrom to 6,200-Angstrom sensitivity; and a 1,500-mm focal-length, narrow-angle camera with a 3,200-Angstrom to 6,200-Angstrom range.

The discs of Jupiter and Saturn exceed the field of view of the narrow-angle camera about three or four weeks before closest approach. At that time, resolution is about 400 kilometers (250 miles). For several days before and after closest approach, scientists will have several simultaneous imaging opportunities:

1. Photography at high resolution of planets whose angular diameters are many times larger than the field of view.
2. Close encounters with the major satellites.
3. More distant photography of several other satellites.

4. High-resolution photography of Saturn's rings.

To exploit such a variety of opportunities, it is necessary for the spacecraft to return large quantities of imaging data. The camera-spacecraft system has been designed to return imaging data over a wide range of telemetry rates in real time. Data can also be recorded on board the spacecraft for later playback to Earth -- during occultation by Saturn, for instance.

Each camera is equipped with a filter wheel whose individual filters have a variety of uses:

The wide-angle camera's filter wheel contains one clear filter, one each in blue, green and orange wavelengths, a seven-Angstrom sodium-D filter for special observations near Io and other satellites, and two 100-Angstrom filters at the wavelength of methane absorption, for study of the distribution of methane in the atmospheres of Jupiter, Saturn, Titan and Uranus.

The narrow angle camera's filter wheel carries two clear filters, two green, and one each of violet, blue, orange, and ultraviolet.

Voyager is the first imaging system with narrow-band capability to directly observe distribution of atomic and molecular species. The seven-Angstrom sodium-D filter is the narrowest bandwidth filter ever flown with this kind of camera.

Because the Voyager spacecraft pass the planets and satellites at high velocities and must take pictures in dimmer light than Mariner missions to the inner planets, image-smear conditions are more severe than on previous flights. To overcome these problems, the camera's pre-amplifiers have been designed to lower system noise and to incorporate a high-gain state; both changes are meant to provide high-quality images with minimum smear.

During the several months before closest approach, the narrow-angle camera photographs Jupiter and Saturn regularly and often to provide information on cloud motions. Those pictures are taken on a schedule that permits scientists to make motion pictures in which the planet's rotation has been "frozen" so that only the cloud motions are apparent. Resolution during the period ranges from about 1,600 kilometers (1,000 miles) to about 400 kilometers (250 miles). Once the planet grows larger than the narrow-angle camera's field of view, the wide-angle camera begins its work. The narrow-angle camera then repeatedly photographs portions of the planets that warrant special scientific interest. The cameras can be shuttered simultaneously during these periods so scientists can relate small-scale motions to larger patterns.

Because of the nature of the planetary flybys, the cameras are not able to concentrate on a single target for hours at a time. As each satellite moves, it presents an ever-changing appearance to the cameras; the planets' clouds are also in constant motion. Therefore, observational sequences are structured to provide repeated images at differing intervals for each target. Additionally, large amounts of multicolor imaging data are being obtained for the planets and satellites.

The camera system weighs 38.17 kilograms (84.15 pounds), and uses 41.9 watts of power.

Dr. Bradford A. Smith of the University of Arizona is team leader.

ULTRAVIOLET SPECTROMETER

The ultraviolet spectrometer looks at the planets' atmospheres and at interplanetary space.

Scientific objectives of the experiment are:

1. To determine distributions of the major constituents of the upper atmospheres of Jupiter, Saturn and Titan as a function of altitude.
2. To measure absorption of the Sun's ultraviolet radiation by the upper atmospheres as the Sun is occulted by Jupiter, Saturn and Titan.
3. To measure ultraviolet airglow emissions of the atmospheres from the bright disks of the three bodies, their bright limbs, terminators and dark sides.
4. Determine distribution and ratio of hydrogen and helium in interplanetary and interstellar space.

The instrument measures ultraviolet radiation in 1,200-Angstrom bandwidth in the range from 400 to 1,800 Angstroms. It uses a grating spectrometer with a microchannel plate electron multiplier and a 128-channel anode array. A fixed-position mirror reflects sunlight into the instrument during occultation. The instrument has a 0.86-degree by 0.6-degree field of view during occultation and a 0.86 by 2 degree field of view for airglow measurements.

The ultraviolet spectrometer weighs 4.49 kilograms (9.89 pounds) and uses 2.5 watts of power.

Dr. A. Lyle Broadfoot of the University of Southern California is principal investigator.

TRACKING AND DATA ACQUISITION

Tracking, commanding and obtaining data from the spacecraft are part of the mission assigned to the Jet Propulsion Laboratory. These tasks cover all phases of the flight, including telemetry from launch vehicle and spacecraft, metric data on both launch vehicle and Voyager, command signals to the spacecraft and delivery of data to the Mission Control and Computing Center (MCCC) at JPL.

The Tracking and Data System (TDS) provides elements of the world-wide NASA/JPL Deep Space Network (DSN), Air Force Eastern Test Range (AFETR), the NASA Spaceflight Tracking and Data Network (STDN) and the NASA Communications System (NASCOM).

During the launch phase of the mission, data acquisition was accomplished through use of the near-Earth facilities -- the AFETR stations, downrange elements of the STDN, instrumented jet aircraft and an instrumented ship. Radar-metric data obtained immediately after liftoff and through the near-Earth phase was delivered to and computed at the AFETR Real-Time Computer system facility in Florida so that accurate predictions could be transmitted to Deep Space Network stations giving the locations of the spacecraft in the sky when they appeared on the horizon.

Tracking and communication with the Voyagers since injection onto Jupiter trajectories and until the end of the mission are being carried out by the Deep Space Network (DSN).

The DSN consists of nine deep space communications stations on three continents, a spacecraft monitoring station in Florida, the Network Operations Control Center at JPL and ground communications linking all locations.

DSN stations are located around the Earth -- at Goldstone, California; Madrid, Spain; and at Canberra, Australia. Each location is equipped with a 64-meter (210-foot) diameter antenna station, a 34-meter (112-foot) antenna, and a 26-meter (85-foot) antenna station.

The three multi-station complexes are spaced at widely separated longitudes so that spacecraft beyond Earth orbit -- and, for the Voyager mission, the planets Jupiter and Saturn -- are never out of view. The spacecraft monitoring equipment in the STDN station at Merritt Island, Florida, covered the pre-launch and launch phases of the mission. A simulated DSN station at JPL, called CTA-21, provided pre-launch compatibility support.

In addition to the giant antennas, each of the stations is equipped with transmitting, receiving, data handling and inter-station communication equipment. The downlink radio frequency system includes super-cooled low-noise amplifiers. The 64-meter antenna stations at Goldstone, Spain and Australia have 100-kilowatt transmitters. Transmitter power at 34- and 26-meter stations is 20 kilowatts.

The downlink is transmitted from the spacecraft at S-band (2,295 megahertz) and X-band (8,400 mhz) frequencies. The uplink operates at S-band (2,113 mhz) only, carrying commands and ranging signals from ground stations to the spacecraft.

Only the 64-meter and 34-meter antenna stations can receive the X-band signal and can receive at both frequencies (S-band and X-band) simultaneously. During Saturn encounter operations, the 64-meter and 34-meter stations provide continuous spacecraft coverage. From about four weeks before to one week after closest approach the arraying of co-located 64-meter and 34-meter antennas is used at all three locations to maximize data-return capability.

During cruise operations a combination of 64-meter, 34-meter and 26-meter station coverage is used for mission operations support. The 26-meter subnet is used when only S-band data are received.

The nerve center of the DSN is the Network Operations Control Center at JPL that provides for control and monitoring of DSN performance. All incoming data are validated at that point while being simultaneously transferred to the computing facilities of the Mission Control and Computing Center for real-time and later use by engineers and science investigators.

Ground communications facilities used by the DSN to link the global stations with the control center are part of a larger network, NASCOM, that connects all of NASA's stations around the world. Data from the spacecraft are transmitted over high-speed and wide-band circuits. For the Saturn encounter, telemetry at rates up to 44.8 kilobits per second are carried in real-time on wideband lines from Goldstone, Canberra and Madrid.

Simultaneously with the routing to the MCCC of the spacecraft telemetry, range and range-rate information is generated by the DSN and transmitted to the control center for spacecraft navigation. To achieve the desired maneuver and encounter accuracies, very precise navigation data are required. Navigation information includes S/X ranging, two-station near-simultaneous ranging, differential time delay between spacecraft and quasar, and multi-station tracking cycles.

Commands are sent from the MCCC to one of the DSN stations where they are loaded into a command processing computer, automatically verified for accuracy and transmitted to the proper spacecraft at 16 bits per second. Commands may be aborted, if necessary.

For all of NASA's unmanned missions in deep space, the DSN provides the tracking information to determine the trajectory of the spacecraft. It receives engineering and science telemetry and sends commands for spacecraft operations on a multi-mission basis.

Concurrent with the four-year or longer Voyager mission, the network is supporting mission activities of the Viking project with one lander on Mars; maintaining post-Jupiter and Saturn communications with Pioneers 10 and 11; and complementing West Germany's space communications facilities on the Helios Sun-orbiting mission. The DSN also is supporting a Venus exploration mission by a Pioneer spacecraft -- a planetary orbiter that began planetary-science activities in December, 1978.

All of NASA's networks are under the direction of the

Office of Space Tracking and Data Systems. JPL manages the DSN. The STDN facilities and NASCOM are managed by NASA's Goddard Space Flight Center, Greenbelt, Maryland.

The Goldstone DSN stations are operated and maintained by JPL with the assistance of the Bendix Field Engineering Corp. The Canberra stations are operated by the Australian Department of Supply. The stations near Madrid are operated by the Spanish government's Instituto Nacional de Tecnica Aerospacial.

MISSION CONTROL AND COMPUTING CENTER

The Mission Control and Computing Center (MCCC) at the Jet Propulsion Laboratory is the focus of all Voyager Project flight operations. It is through the center's computer systems that data from the Voyagers pass, are processed and presented to engineers and scientists for analysis. Through the extensive and varied displays of the computers in the MCCC, the flight analysts observe and control the many ground processing functions and the spacecraft.

The MCCC is housed in JPL buildings containing its computer systems, communication and display equipment, photo processing lab and mission support areas. The various areas are outfitted to satisfy the diverse needs of the Voyager mission operations team -- the requirements of the mission controllers, spacecraft performance analysts and science investigators.

The MCCC contains several computer systems designed to receive the incoming Voyager data, process it in real-time, display it and organize it for further processing and analysis. After the data have been received as radio signals by the Deep Space Network (DSN) stations located around the world, they are transmitted to Pasadena and into the MCCC computers where the processing is done. Software developed by the MCCC, operating in those computers, performs the receiving, displaying and organizing functions. Computer programs generated by other elements of the Voyager Project further process the data.

Commands causing the spacecraft to maneuver, gather science data and perform other complex mission activities are introduced into the MCCC computers and communicated to a station of the Deep Space Network for transmission to the appropriate spacecraft.

The MCCC is composed of three major elements, each with its own computer system. They are the Mission Control and Computing Facility (MCCF), the Information Processing Center (IPC) and the Mission and Test Computing Facility (MTCF).

The Mission Control and Computing Facility consists of two IBM 360-75 processors and a number of Modcomp computers. It supports the Voyager command, monitor and data records activities and routes data to the MTCF and tracking systems. The Modcomp computers provide the means by which commands are formatted and sent by the Deep Space stations to the spacecraft. They also provide displays of ground data system configuration and status information. The 360-75s are used to provide the data management capability to produce plots and printouts for the day-to-day determination of spacecraft operating condition. The 360-75s also produce the final records of data for detailed analysis by the science community.

The Information Processing Center, with two UNIVAC 1100-81 computers, supports the Voyager Project's navigation, spacecraft analysis, and mission sequence systems. Computer terminals located in the mission support areas allow project analysts to execute their programs and obtain results displayed on TV monitors, or on various printers and plotters.

The Mission and Test Computing Facility provides telemetry data processing for the science and engineering information transmitted from the Voyagers. Within the MTCF are the telemetry system, imaging system and photo system. The telemetry system uses three strings of UNIVAC and Modcomp computers to receive, record, process and display the data as requested by analysts in the mission support areas. The imaging and photo systems produce the photographic products from data generated by Voyager's TV cameras. Pictures of Jupiter, Saturn and their moons are analyzed by scientists housed in the mission support areas. Scientists are provided both electronic and photographic displays.

MCCC, like the DSN, also supports the other flight missions, Viking, Pioneers 10 and 11, Helios and Pioneer/Venus.

VOYAGER SUBCONTRACTORS

This is a list of some key subcontractors who provided instruments, hardware and services for the Voyager Project:

Algorex Data Corp. Syosset, NY	Automated Design Support for Flight Data Subsystem
Boeing Co. Seattle, WA	Radiation Characterization of Parts and Materials
Fairchild Space and Electronics Co. Germantown, MD	Temperature Control Louvers
Ford Aerospace and Communications Corp. Palo Alto, CA	S/X Band Antenna Subsystem; Solid State Amplifiers
Frequency Electronics, Inc. New Hyde Park, NY	Ultra Stable Oscillators
General Electric Co. Space Division Philadelphia, PA	Radioisotope Thermoelectric Generators
General Electric Co. Utica, NY	Computer Command Subsystem; Flight Control Processors
General Electric Co. Space Systems Organization Valley Forge, PA	Attitude Control and Articulation Subsystem
Hi-Shear Corp. Ordnance Division Torrance, CA	Pyrotechnic Squibs
Honeywell, Inc. Lexington, MA	Canopus Star Trackers
Hughes Aircraft Co. Aerospace Group Culver City, CA	Radiation Characterization of Parts and Materials
Lockheed Electronics Co. Industrial Technology Div. Plainfield, NJ	Data Storage Tape Transport

Martin Marietta Aerospace Denver, CO	Attitude Control Electronics; Propulsion Subsystem
Motorola, Inc. Government Electronics Div. Scottsdale, AZ	Modulation/Demodulation Subsystem; Radio Frequency Subsystem
Rocket Research Corp. Redmond, WA	Rocket Engine and Thruster Valve Assemblies
SCI Systems, Inc. Huntsville, AL	Computer Command Subsystem Memories
Teledyne Microelectronics Los Angeles, CA	Hybrid Memories for Flight Data Subsystem
Texas Instruments Dallas, TX	Data Storage Electronics
The Singer Company Little Falls, NJ	Dry Inertial Reference Units (Gyroscopes)
Thiokol Chemical Corp. Elkton Division Elkton, MD	Solid Rocket Motor
Watkins-Johnson Co. Palo Alto, CA	S/X Band Traveling Wave Tube Amplifiers
Xerox Corp. Electro-Optical Systems Pasadena, CA	Power Subsystem
Yardney Electronics Corp. Denver, CO	Flight and Test Battery Assemblies
Science Instruments	
Massachusetts Institute of Technology Cambridge, MA	Plasma Subsystem
University of Colorado Boulder, CO	Photopolarimeter Subsystem
University of Iowa Iowa City, IA	Plasma Wave Subsystem
Xerox Corp. Electro-Optical Systems Pasadena, CA	Imaging Science (TV) Electronics
Kitt Peak National Observatory Tucson, AZ	Ultraviolet Spectrometer

Johns Hopkins University Applied Physics Laboratory Baltimore, MD	Low Energy Charged Particle Subsystem
Goddard Space Flight Center Greenbelt, MD	Magnetometers; Cosmic Ray Subsystem
Texas Instruments Dallas, TX	Modified Infrared Interferometer Spectrometer and Radiometer
Martin Marietta Aerospace Denver, CO	Planetary Radio Astronomy Subsystem
Astro Research Corp. Santa Barbara, CA	Magnetometer Boom; Planetary Radio Astronomy Antennas
TRW Defense and Space Systems Redondo Beach, CA	Ultraviolet Spectrometer Electronics
Matrix Corp. Acton, MA	Plasma Subsystem Electronics
General Electrodynamics Corp. Dallas, TX	TV Vidicons

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APPENDIX A: EARTH-SPACECRAFT COMMUNICATIONS

The communication process between Voyager 2 and Earth involves several unique features. Transmission of commands is constrained by failure of the prime radio receiver and partial failure of that portion of the back-up receiver used to aquire and track the transmitted signal. Several special actions have been taken to accommodate the limitation. Data transmission to Earth takes advantage of the benefits of increasing apparent size of the receiving antenna by arraying together two antennas, but is complicated by the effects of terrestial weather.

Earth to Spacecraft - - Uplink

Voyager was launched with two phase-lock-loop radio receivers. Unlike a car radio receiver, which can be tuned to receive signals from stations transmitting at different frequencies, the Voyager receivers have a single "best-lock frequency" (BLF) and the capability of locking on to and then tracking a received signal over a frequency range of 500,000 Hertz centered on the receivers' BLF. The 500,000 Hertz is called the tracking-loop bandwidth. A tracking-loop bandwidth of at least 150,000 Hertz is required to accommodate the Doppler effects induced by rotation of Earth and acceleration of the spacecraft by the planet, and to accommodate changes in the BLF caused by temperature variations in the receiver. Deep Space Stations (DSS) cause this to happen by slowly increasing or decreasing the transmitted frequency over a portion of the tracking bandwidth to allow the spacecraft receiver to lock onto the transmitted signal. This is called ramping. The station then tunes to a fixed frequency for the remainder of the tracking period, usually

8 to 12 hours. Frequency shifts during the rest of the tracking period due to rotation of Earth, about 6,000 Hertz, or spacecraft acceleration, as high as 82,000 Hertz near Saturn closest aproach, are then accommodated by the large tracking bandwidth.

In April 1978 the Voyager 2 prime receiver failed and the back-up receiver was determined to have a shorted capacitor in the phase-lock-loop filter. The short reduces the tracking-loop bandwidth from 500,000 Hertz to 200 Hertz. The narrowness of the new bandwidth requires that the frequency transmitted from the DSS be constantly tuned to remove the rotating Earth and spacecraft acceleration effects. A variety of ground constraints requires that the frequency received by the spacecraft not vary by more than about 50 Hertz from the BLF. An emergency development of special hardware resulted in the capability to meet the requirement. But there is another complication. The sensitivity of the receiver's voltage-controlled oscillator to small changes in the receiver temperature, approximately 380 Hertz per degree celsius, causes large swings in the BLF, relative to the 200 Hertz bandwidth. Examples of events that induce temperature variations large enough to change the BLF are power changes anywhere on the spacecraft to more than two watts and spacecraft maneuvers that change the solar illumination of the main structure.

The sequence of events for the Voyager 2 Saturn encounter has been designed to reflect the sensitivity of the BLF to temperature variations. A history of the effects on the BLF of various configuration changes and maneuvers has been documented in a Flight Rule. The rule specifies the time (referred to as a "command moratorium" since command loads cannot be transmitted and reliably received by the spacecraft during those periods) that must be set aside to allow the thermal transient resulting from an event to stabilize. Normal command moratoria are between 24 and 72 hours. Near the end of a command moratorium, special procedures are used to measure the new BLF. The same procedures are performed at the beginning of most DSS tracking periods to confirm that no unexpected changes in the BLF have occurred.

Execution of the BLF determination procedures and restrictions on spacecraft activity are important before command load periods and navigation tracking cycles since a command load period is 12 hours long and nav cycles are between 30 and 54 hours long. The frequency of command loads and nav cycles is dictated by spacecraft activity planned for a given time and the need to obtain measurements of the range and velocity of the spacecraft relative to Earth so the trajectory can be determined. Between 25.8 days before Saturn encounter and one day after, six command loads and three nav cycles must be accommodated, having a net of 16.4 days to execute, one trajectory correction maneuver, 14 science maneuvers, and three significant power changes. The command moratoria imposed by those events add up to 8.4 days, leaving a net of eight days of unsubscribed tracking time. In addition to that broad based problem, the day of Saturn closest approach poses two special difficulties. A key science objective is determination of the mass of Tethys. This means the spacecraft must be in lock with the transmitted signal for a period of about 50 minutes, beginning less than five minutes after the completion of the exit

Earth-occultation maneuver performed for radio science. During Earth occultation (when the spacecraft is hidden from Earth) two of the largest power transients of the entire encounter are made to achieve configuration for the radio science maneuvers. The X-band transmitter is reduced to low power and the S-band transmitter is raised to high power for a net power increase of 28 watts, and the telemetry drivers are turned off for a net decrease of approximately one watt. Those changes to the configuration are in effect for 2.5 hours only, ending 15 minutes before the end of the exit Earth-occultation radio science maneuvers; however, they are expected to cause a shift in the BLF of at least 65 Hertz. Normally, these changes would result in a 72-hour command moratorium. The shortness of the period for which the transmitter power changes are in effect suggests that a shortened command moratorium of 12 hours would be sufficient. As a contingency for an error in prognosis, a second load-start point has been planned at 46 hours after Saturn closest approach, or 19 hours after the nominal load-start point. Thus, should it be necessary to use the contingency plan, about 19 hours of science data would be lost.

Spacecraft to Earth - - Downlink

Key elements of the Voyager downlink consist of low temperature receivers at the Deep Space Stations, efficient error-correction codes for formatting the data, and use of X-band frequencies (8,000 megaHertz) to maximize downlink data rates. The low-temperature receivers are crucial to detection of the extremely weak signal (0.000,000,000,000,000 1 watt) striking the DSS antenna.

Most of the data transmitted by the spacecraft is convolutionally coded. Coding the data in that manner results in a probability distribution for bit errors that is very sensitive to the signal-to-noise ratio of the receiver. Specifically a 25 percent change (1 dB) in the signal-to-noise ratio results in a factor-of-10 change in the bit-error rate.

Alternatively, since simultaneous receipt and processing of the received signal from the spacecraft by two co-located stations (one 64-meter and one 34-meter) increases the receiver signal-to-noise ratio by about 1 dB, errors in the data can be reduced by 90 percent by arraying a 64-meter and a 34-meter Deep Space station. For the period between 26 days before encounter to about seven days after closest approach arraying is performed at the DSS complexes to take advantage of the improved performance.

Loss of the 34-meter station during any arrayed track would result in reduction of data quality or quantity, but not in the loss of all data. Loss of the 64-meter antenna would cause loss of X-band capability, reducing the data to engineering measurements only.

The X-band signal is strongly affected by bad weather -- water vapor or rain. Sequences have been designed to accommodate statistical weather variations. On most days downlink performance will be good. Data loss could be extensive on days where heavy rain or snow falls on the antennas. Those losses are not expected to exceed 10 percent of all data. To protect against even those losses -- where data have been defined as unique or critical --

the data are recorded by the spacecraft and played back twice over separate Deep Space Network stations. That provides additional protection against possible bad weather.

APPENDIX B NAVIGATION OPERATIONS

Navigation is the art of determining where the spacecraft is and where it is going, and making corrections to its trajectory so it will arrive at the intended place at the desired time. The process involves three conceptually different functions:

Making measurements that are sensitive to the spacecraft's position or velocity.

Using the observations to correct the prediction of where, with respect to the target planet, the spacecraft is going and when it will arrive at the predicted place. Using the predicted arrival place and time to determine whether the spacecraft will be close enough to the desired aiming conditions to satisfy mission requirements, and if not, computing the trajectory correction maneuver necessary.

Navigation Measurements

Measurements for navigation are of two basic types: radiometric and optical. Radiometric observations are made by a station on Earth using radio signals to and from the spacecraft. Optical observations are pictures taken by the camera of the target body against the background of stars.

Radiometric observations are obtained by tracking the spacecraft with the antennas of the Deep Space Network, strategically located at different longitudes on the Earth so that it is possible to have the spacecraft in view at any time on a given day. The tracking stations measure two quantities of interest to the navigator: The difference in frequency between the signals sent to the spacecraft and those received from it, caused by relative motion between spacecraft and station. That difference is called Doppler shift. It is the same effect that causes a train whistle to change in pitch as it passes an observer, and it provides the navigator with a precise measurement of the velocity of the spacecraft along the line of sight from the station. The stations also measure the time required for a signal to travel to the spacecraft and back. That is called range measurement because it is directly related to the distance (range) between the station and the spacecraft.

Radiometric data is used for navigation during all mission phases. However, during encounters optical data from the cameras complements radio data and becomes the navigator's most powerful tool. It has the advantage of measuring the spacecraft's position relative to the target rather than to Earth and the further advantage of becoming more accurate as the spacecraft gets closer to its target.

Of approximately 17,500 pictures expected to be taken by Voyager 2 during its Saturn encounter, about 180 are designed for optical navigation. Those pictures use the narrow-angle camera for high resolution and are planned to capture a star image in the same field of view with one or more of Saturn's moons. Pictures of the moons are used rather than Saturn itself because Saturn's image would fill the narrow-angle field of view 16 days before Saturn encounter and would be too large to reliably capture with a star much earlier than that. The star provides a precise direction reference in the picture from which the satellite positions can be measured and the spacecraft's position can be inferred.

Orbit Determination

Turning raw measurements into predicted encounter conditions is called orbit determination (OD in navigator's jargon). Orbit determination is a differential correction process. The OD analyst starts with a reference trajectory, usually his previous best estimate, and computes a correction based on the difference between what was observed in the navigation measurements and what mathematical formulas predict should have been observed with the current trajectory.

Accuracy of the estimated trajectory depends on the accuracy of the measurements and on the accuracy and completeness of the mathematical models embodied in computer programs that represent everything known to significantly affect spacecraft motion or the measurements. Uncertainties in such diverse quantities as position of the target planet in the solar system, locations of tracking stations on Earth's crust, wobble of the Earth about its spin axis, solar pressure and attitude control activity on the spacecraft, charged particles in space that delay the radio signals, and electromagnetic distortion of images from the spacecraft cameras limit the accuracy achievable.

Despite the diversity (and perversity) of error sources affecting orbit determination, the Voyager spacecraft can be navigated using only radiometric data to an accuracy slightly better than one part in a million: At a distance of 1 million miles, navigation error would be less than a mile. That is comparaole to hitting a target the size of a baseball from 25 miles away. Unfortunately, Saturn is 968 million miles from Earth at the time of Voyager 2's encounter, so the corresponding radiometric accuracy is about 800 miles. That is why optical data is so important for Voyager navigation. It enables navigators to reduce the expected delivery error relative to Saturn by a factor of 15 compared with what would be achievable with only Earth-based tracking. Instead of 800 miles the expected error with both radio and optical data is 55 miles. That is comparable to hitting a target the size of an aspirin tablet (rather than a baseball) 25 miles away.

Since optical data continues to provide more accurate measurements as the spacecraft gets closer to Saturn, trajectory knowledge continues to improve even after the final trajectory correction maneuver has been executed. The additional trajectory knowledge cannot affect the flight path, but is very important for ensuring that the science instruments are accurately pointed at their intended targets during the time within a few hours of closest approach. Getting correct instrument pointing will be most difficult for objects such as Tethys and Enceladus that are viewed from close range (less than 100,000 km) and for objects whose positions are less well known than the spacecraft's. In this category are several of the recently discovered small satellites for which relatively close-up imaging will be attempted by Voyager 2. One of the navigation challenges for this encounter is the "satellite pointing update", which begins 2 1/2 days before Saturn closest approach, during which improved orbit knowledge for the satellites as well as the spacecraft will be exploited to make final corrections to the instrument pointing for selected key observations.

When the excitement of the encounter has waned, there is one more orbit determination task to be done: The encounter trajectory reconstruction. "Reconstruction" refers to the precise determination, after the fact, of what trajectory was actually flown through the encounter. That is essential for interpretation of science data from the encounter, especially for some of the radio-science investigations. The post-encounter trajectory reconstruction error (relative to Saturn) will be only 2% of the expected delivery error: The reconstructed trajectory will be accurate to about 1 mile. Ironically, it is not the optical data that makes that accuracy possible but the radiometric data -- the Doppler measurements. That is because Saturn's gravity causes the trajectory to bend during the flyby, and the bending induces large velocity changes that are precisely measured by the Doppler. A side benefit of the reconstruction is that it enables navigators to evaluate their performance with respect to the pre-encounter orbit determination.

Trajectory Correction Maneuvers

Trajectory correction maneuvers (TCMs) control the flight path by changing the spacecraft's velocity slightly, either speeding it up or slowing it down, and changing its direction. TCMs are accomplished by four small thrusters on the spacecraft opposite the large high-gain antenna. To execute a TCM the spacecraft is first turned so the thrusters point in the right direction to effect the desired velocity change. Then they are fired. The four thrusters acting together exert a maximum force of 1/2 pound on the spacecraft. Length of the burn determines magnitude of the velocity change. A velocity change of 1 mile an hour requires a burn of about 3 minutes. A typical TCM changes the velocity by only a few miles an hour. Since the spacecraft travels through the solar system at 35,000 mph (relative to the sun), a typical correction would represent a change of 1 part in 10,000 or .01%.

The last TCM executed by Voyager 2 was TCMB7 (the seventh TCM for the "B" spacecraft) on Feb. 26, 1981. The final approach to Saturn will be controlled by two additional TCMs (B8 and B9) on July 19 (Saturn encounter minus 37 days) and Aug. 18 (encounter minus 7 days). Each maneuver is expected to change the velocity by 1 to 5 mph. Their placement was chosen to achieve the necessary delivery accuracy with minimal propellant.

The engineering trade-off between accuracy and propellant use is a major consideration in planning the trajectory correction strategy. The accuracy to which the spacecraft is delivered to its target is determined by the final pre-encounter TCM, which in turn is limited by the orbit determination accuracy available at the time. Since OD accuracy improves as the planet is approached, a late TCM provides more accurate delivery. On the other hand, the velocity change required for a given correction in encounter position is inversely proportional to the time-to-go. The closer the spacecraft is to Saturn when the TCM is executed, the more propellant is required to make the correction. For that reason two TCMs control the encounter trajectory rather than one. TCMB8 will remove most trajectory errors accumulated during previous months of cruise when little propellant is required. Then TCMB9 will take advantage of the improved OD accuracy to make a final small correction. That two-maneuver approach uses only about 40% of the propellant required if a single large correction were made at the time of TCMB9. Conservation of propellant is important for Voyager 2, as its flight to Uranus and Neptune depends on sufficient fuel beyond Saturn.

Targeting Strategy

The aim point at Saturn was selected to continue the "Grand Tour" trajectory that Voyager 2 is following through the solar system. That targeting strategy uses the gravitational field of each planet to "sling" the spacecraft on to the next planet. Voyager 2 gains energy with each planetary encounter. Had there been no planets to encounter the spacecraft would have stayed in orbit around the Sun. Instead, using energy from the planets, it will leave the solar system at 36,000 miles an hour.

Each encounter is carefully targeted to send Voyager 2 to the desired position at the next planet, but a TCM is needed to adjust the timing of the next encounter. On Sept. 29 a post-Saturn TCM (called B10) is planned to adjust the arrival time at Uranus by two days to meet the Uranus satellites Miranda and Ariel. TCMB10 will speed up the trajectory by 34 miles an hour and will correct the direction of flight to compensate for the delivery error at Saturn. There is a 4,500-to-1 multiplier between delivery error at Saturn and the error at Uranus: For every mile that Voyager 2 misses its Saturn aimpoint the resulting flight path (if not corrected) would miss Uranus by 4,500 miles. Thus, the magnitude of the velocity correction to be made by TCMB10 (and therefore the propellant expenditure) is directly related to the Saturn-relative delivery error. That further emphasizes the importance of optical data in enhancing the orbit determination accuracy. Consider that without optical data there would be a 90% chance of having enough fuel to reach the desired Uranus encounter and only a 10% chance of reaching Neptune. With optical navigation the fuel for the Uranus encounter is assured, and the probability of reaching Neptune is 99%.

Navigation Status

The figure shows the navigation status in the "aim plane". (Navigators call it the "B-plane"). The aim plane is an imaginary plane passing through the center of the target planet that represents the desired encounter conditions and orbit determination results. Included are the aim point and the current best estimate (CBE) for the encounter trajectory based on orbit determination of May 15, 1981. Associated with each point in the aim plane is an arrival time expressed in hours, minutes, and seconds GMT on Aug. 26, 1981. The error ellipse defines the uncertainty associated with the CBE. Statistically, there is a 40% probability that the actual trajectory is within the error ellipse associated with the orbit determination estimate. That information will be updated during the Saturn enounter.

Encounter Trajectory Description

The Voyager 2 encounter differs in major respects from the Voyager 1 Saturn encounter. Since Saturn was scheduled to be the final planetary encounter for Voyager 1, the trajectory could be designed to maximize the science return and safety of the spacecraft without regard to its post-Saturn characteristics. For Voyager 2, the aim point was defined solely by the requirements to continue the trajectory to Uranus. The arrival time, however, was selected to provide close encounters with some satellites that were not seen close-up by Voyager 1 -- Tethys and Enceladus. Closer approaches to Hyperion, Iapetus, and Phoebe are also achieved by Voyager 2, although none is really close. Other differences include a slightly lower-altitude flyby of Saturn and the absence of a close Titan encounter, one of the major objectives of Voyager 1. Voyager 2 crosses the potentially hazardous ring plane only once rather than twice, but it does so at a point much closer to Saturn, 2.85 Saturn radii rather than 6.26. A general view of the Voyager 2 Saturn encounter from above the trajectory plane is shown in the figure. Voyager 2 encounters Saturn on Aug. 25, 1981, at a nominal spacecraft event time of 8:24 p.m. PDT. Closest approach will be 161,094 km or 2.67 Saturn radii from the planet center, which corresponds to an altitude of 101,000 km (63,000 mi) above the cloud tops.

Prior to closest approach, the spacecraft encounters five major satellites: Iapetus (-74 hrs, 909,000 km), Hyperion (-26 hrs, 471,000km), Titan (-18 hrs, 666,000 km), Dione (-2 hrs, 502,000 km), and Mimas (-1 hrs, 310,000 km). The spacecraft also encounters four minor satellites or "rocks:" S12 (-4 hrs, 318,000 km) S15 (-2 hrs, 313,000 km*), S17 (-1 hr, 154,000 km), and S13 (-5 min, 107,000 km). In addition to the above satellite encounters, the star occultation of Delta-Scorpius (Dschubba) by Saturn and its rings occurs before the spacecraft reaches periapsis.

After Saturn closest approach, the spacecraft flies by the four remaining major satellites: Enceladus (+21 min, 87,000 km), Tethys (+3 hrs, 93,000 km), Rhea (+3 hrs, 645,000 km) and

Phoebe (+10 days, 2,076,000 km). Closest approaches to three more rocks also occur after periapsis: S14 (+9 min, 247,000 km), S10 (+30 min, 223,000 km); and S11 (+42 min, 147,000 km). The trajectory descending node (ring plane crossing) is reached about 54 minutes after closest approach to Saturn, at a range of 172,109 km or 2.853 Saturn radii.

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* Closest viewing distance. S15 is occulted by Saturn at closest approach.